AN INVESTIGATION OF THE EFFECTS OF THE SUDDEN EXTENSION OF A DIVE-RECOVERY FLAP ON THE AERODYNAMIC CHARACTERISTICS OF A SYMMETRICAL AIRFOIL IN TWO DIMENSIONAL FLOW

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SYMBOLS

M = Free Stream Mach Number

α = Absolute Angle of Attack

C_L = Lift Coefficient

$C_{L_{Crit}}$ = The maximum lift coefficient obtainable for a given angle of attack with increasing Mach number

$$\Delta C_{L_{Crit}} = (C_{L_{Crit}})_{\text{Flap popped}} - (C_{L_{Crit}})_{\text{Flap set}}$$

p = Free Stream Static Pressure

p_o = Reservoir Static Pressure

$\gamma = C_p/C_v = 1.40$ (Air)

SUMMARY

This report presents the results of an investigation of the aerodynamic characteristics of the NACA symmetrical laminar flow 65,1-012 airfoil. The model was tested with and without a dive recovery flap. The effects obtained by suddenly extending the flap are compared to those obtained with the flap set down in position. The tests cover a range of angle of attack from zero to plus three degrees while the Mach number was varied from 0.5 to 0.8.

The following conclusions were reached:

1. The lift curve for the airfoil with no flap is essentially a straight line.

2. A flap suddenly extended produces aerodynamic effects which are different from those produced by a flap which is set. Whenever model test results are to be used to produce design information to be incorporated in full scale aircraft, the dive recovery flaps on the model should be equivalent to those on the full scale airplane in regard to dimensional proportions and to the method and timing of operation, if a high degree of accuracy is desired.

This investigation was carried out by the authors at the Guggenheim Aeronautical Laboratory of the California Institute of Technology during the school year 1945-46.

INTRODUCTION

The primary purpose of this investigation is to make a comparison of the two-dimensional aerodynamic effects of a dive recovery flap set in the extended position with the effects of suddenly extending or popping the flap at a given Mach number. In all known previous wind tunnel tests involving dive recovery flaps the flap has been set on the model in the extended position, and the free stream velocity built up to the test Mach number. In this investigation it is attempted to determine if the latter procedure may produce misleading results inasmuch as the technique does not simulate the manner in which dive recovery flaps are actually used in flight.

The secondary purpose of this investigation is to determine the lift characteristics of the NACA 65,1-012 airfoil in two-dimensional flow. This work is an extension of the results of a previous investigation carried out on the same model in the same wind tunnel, Ref. 1.

The tests covering the effects of the dive recovery flaps were made with the NACA symmetrical, low-drag section, 65,1-012, with a four inch chord. The study was limited to tests with one flap size at three chordwise locations, in the set and in the popped conditions. Angle of attack was varied from zero to plus three degrees, and Mach number was varied from about 0.5 to 0.8 for the flap tests and for the investigation of the lift characteristics of the model with no flaps. The Reynolds number corresponding to above Mach numbers was of the order of 1.6×10^6.

[illegible]

Before undertaking the major part of this project concerning the dive recovery flap, it was desired to first check the accuracy of the results obtained in this tunnel by previous investigators. These results will be found in the report listed as Ref. (1). Since in that report the curves of C_L versus α do not pass through $\alpha = 0$ at $C_L = 0$, an effort was made to determine whether the error was attributable to inaccuracy in setting the angles of attack, or in measuring the lift. This was necessary in order to eliminate the possibility of obtaining similar erroneous results. It was for this reason that a new system of setting angle of attack was devised (described under EQUIPMENT). Since if it were known that the angle of attack were correct to within 0.03 of a degree, the error, if any, could only lie in measurement of lift.

The model was installed in the tunnel and set as closely to an angle of attack of zero degrees as possible by lining up the axis of the airfoil parallel to the axis of the test section. The mirror was then fixed on the end of the airfoil shaft at the position which reflected zero degrees on the angle of attack scale as viewed through the theodolite. Runs were then made at a Mach number of 0.6, for a range of angles of attack between 0 degrees and +5 degrees as measured by the theodolite, and a curve of C_L versus α plotted.

Since it was known from the geometry of the angle of attack system that the largest error to be expected was 0.03 degrees in any

EQUIPMENT

The tunnel used throughout this investigation is a closed throat, open return, induction tunnel (Fig. 1). A compressor system comprised of two electrically driven, rotary, positive displacement compressors operating in series supplies the air to a system of jets located in a rectangular section downstream of the test section. These jets discharge into a constant area mixing section which is followed by an expanding section exhausting into the ventilation system of the building. The maximum pressure of 100 $/in^2$ which the compressors can deliver limits the speed of the tunnel to a Mach number of 0.87 with no model installed. The use of a remotely controlled, electrical, by-pass valve affords a means for obtaining any desired Mach number below 0.87.

The test section is 20 in. in length and rectangular in cross section 0.9 in. wide, 10.0 in. high at the entrance 1.0 in. wide, 10.0 in. high at the exit. This taper in width is necessary to allow for boundary layer growth. The top and bottom walls are made from machined brass each containing 23 static pressure orifices. These are spaced 0.5 in. apart in the vicinity of the model and 1.0 in. apart elsewhere, for the purpose of obtaining the pressure distribution at these walls. Securely gasketed to the brass upper and lower walls are two 0.625 in. glass plates which form the side walls.

Three models of the NACA 65,1-012 low drag section were used in this investigation. These were machined from brass to a tolerance

of 0.001 in. Each model is of 4 in. chord and contains a retractable dive-recovery flap. On the first model, the flap leading edge in the extended position is located at 15% chord, the second at 30% chord, and the third at 45% chord. The flap on each of the three models is identical. It extends the width of the model and the flap chord is 10% of the airfoil chord. In the extended (popped) position the flap is deflected 45 degrees from the centerline of the airfoil. In the retracted position, it is recessed so that it is completely flush with the lower surface.

It is possible to extend the flap at any time during a run by means of a wire attached to the flap mechanism within the airfoil. This wire is led out of the airfoil through the trailing edge, downstream and out of the tunnel wall through a small hole, and then attached to a simple clamping device. To pop the flap it is only necessary to pull the wire until the flap reaches its limiting deflection of 45 degrees and then clamp the wire to hold the flap securely in position. To retract the flap, the wire is slacked and the flap is pushed up to its recessed position. For details of the flap mechanism see Figs. 2A and 2B.

The model is supported at the 25% chord point in the center of the test section by a through shaft which is rigidly fixed to the airfoil by a set screw countersunk into the side of the airfoil. This shaft extends through small brass trunnions inserted in holes drilled in the glass plates. Gaskets on the sides of the model provide a snug fit with the glass plates and help to uniformly distribute the

pressure which the glass plates exert. This securely holds the model at any desired angle of attack.

An optical system for setting angle of attack was devised in order to obtain maximum accuracy in this respect. This consisted of a theodolite, a small mirror attached by set screw to the end of the airfoil shaft, and a suitable scale of angle of attack graduated in degrees. The theodolite is located 142 in. from the mirror and the scale is located above the theodolite 120 in. from the mirror. See Fig. 3. By sighting through the theodolite at the reflection of the angle of attack scale in the mirror, the position of the horizontal cross hair of the theodolite with respect to the reflected image of the angle of attack scale could be easily established.

Due to the fixity between mirror, shaft, and airfoil, a change of angle of attack of the airfoil produces a corresponding angular deflection of the mirror and consequently a change in the reflected portion of the angle of attack scale as seen through the theodolite. By this system a change in angle of attack $\Delta\alpha$ could be determined to an accuracy of 0.03 of a degree.

The static pressure orifices in the top and bottom walls were connected to a conventional closed system multiple tube manometer containing butyl alcohol for operation at low Mach numbers, and mercury for operation at high Mach numbers. The reference pressure was connected to an orifice near the entrance of the test section at which the pressure was the least affected by the presence of the model. This orifice was also connected to a mercury manometer which was vented to

the atmosphere for purposes of determining free stream pressure. Using the relationship

$$\frac{p}{p_0} = \left(1 + \frac{\gamma - 1}{2} M^2\right)^{-\frac{\gamma}{\gamma - 1}}$$

a scale was constructed so that Mach number could be read directly from this manometer.

Visual studies of the various flow conditions and the photographic prints in the back of this report were made possible through the use of conventional schlieren apparatus, (see Fig. 4). An incandescent light source employing a projection type bulb was used for visual observations. A high intensity spark giving an exposure time of the order of 10^{-4} seconds was used for photographic purposes.

PROCEDURE

Before undertaking the major part of this project concerning the dive recovery flap, it was desired to first check the accuracy of the results obtained in this tunnel by previous investigators. These results will be found in the report listed as Ref. (1). Since in that report the curves of C_L versus α do not pass through $\alpha = 0$ at $C_L = 0$, an effort was made to determine whether the error was attributable to inaccuracy in setting the angles of attack, or in measuring the lift. This was necessary in order to eliminate the possibility of obtaining similar erroneous results. It was for this reason that a new system of setting angle of attack was devised (described under EQUIPMENT). Since if it were known that the angle of attack were correct to within 0.03 of a degree, the error, if any, could only lie in measurement of lift.

The model was installed in the tunnel and set as closely to an angle of attack of zero degrees as possible by lining up the axis of the airfoil parallel to the axis of the test section. The mirror was then fixed on the end of the airfoil shaft at the position which reflected zero degrees on the angle of attack scale as viewed through the theodolite. Runs were then made at a Mach number of 0.3, for a range of angles of attack between 0 degrees and +5 degrees as measured by the theodolite, and a curve of C_L versus α plotted.

Since it was known from the geometry of the angle of attack system that the largest error to be expected was 0.03 degrees in any

value of $\Delta\alpha$, the intersection of the C_L versus α curve with the line $C_L = 0$ gave the exact correction to apply to the angle of attack as read from the scale by the theodolite in order to determine the true α. In this manner the discrepancies in Ref. (1) were determined. (see RESULTS and DISCUSSION).

The runs necessary to compile the data on the effects of the sudden extension of the dive recovery flap were then undertaken. The first model (i.e. flap at 15% chord) was set at an angle of attack of zero degrees and two sets of runs were made at various Mach numbers from M = 0.5 up to a value of Mach number at which the tunnel became choked. The first set of runs was made popping the flap; the second set made leaving the flap extended and then bringing the tunnel up to the desired speed.

In the first case, the tunnel was brought up to speed with the flap retracted. The flap was then popped and the Mach number recorded after the flow had become steady. Since the free stream Mach number decreased after popping the flap, the value of the Mach number after the flap was popped was the required value. Then to get comparative results, the second set of runs with flap extended was made at the same Mach numbers as were obtained after popping in the first set of runs.

This same procedure was repeated for each model at three angles of attack between 0 degree and +3 degrees.

Schlieren photographs were taken on all runs in order to obtain qualitative results in comparing the effects of the sudden extension of the flap as opposed to the effects caused by having the flap initially set.

To determine the lift on model and hence C_L the following procedure was employed. A large sheet of cross section paper was placed behind the glass tubes of the multitube manometer board so that for any run the fluid level in the tubes could be marked on this sheet. From these marks the difference in pressure Δp between any pair of orifices (top and bottom) could be determined. These Δp's were plotted versus tunnel length. A determination of the area under this curve by planimeter afforded means of calculating the lift and hence C_L. These values of C_L and the corresponding Mach numbers were then corrected by the methods outlined in Ref. 2.

RESULTS AND DISCUSSION

The principal physical property measured in this investigation was the lift coefficient of the airfoil model. The values of the lift coefficient with no flap and with various flap conditions were determined from corrected test data and plotted against angle of attack and Mach number, see Fig. 5 to 11 inclusive. Fig. 12 was constructed by taking the $C_{L\,crit.}$ at each angle of attack and at various flap locations (from Fig. 9 - 11 inclusive). Then these $C_{L_{crit.}}$ values were plotted against angle of attack for various flap positions. $C_{L_{crit.}}$ is defined as the maximum lift coefficient obtainable for a given angle of attack with increasing Mach number. Fig. 13 was obtained by taking the mean difference between $C_{L_{crit.}}$, flaps popped and $C_{L_{crit.}}$, flaps set, thus a $\Delta C_{L_{crit.}}$ was obtained, where $\Delta C_{L_{crit.}} = (C_{L_{crit}})_{popped} - (C_{L_{crit.}})_{set}$. $\Delta C_{L_{crit.}}$ was then plotted against flap position in percent of chord. The schlieren photographs taken were grouped according to chordwise flap location, see Fig. 14 - 16 inclusive.

Since this investigation covers essentially two problems, the results of the two parts are discussed separately.

In correlating the results of the secondary problem, namely, the determination of the lift characteristics of the two-dimensional airfoil, the results are compared to those of a previous investigation which employed the same wind tunnel and airfoil model, Ref. 1. In the Results and Discussion of Ref. 1, p. 1.12, the authors stated that their "values of C_L did not lie on a straight line themselves, nor

did any straight line passing through the origin appear to approximately satisfy the points". A review of their plot of C_L vs. α (Ref. 1, Fig. 70) showed that their values of C_L do approximate a straight line and intersect the zero lift axis at approximately the same point, about -0.6 degrees, see Fig. 8. We know from two dimensional airfoil theory and tests that the lift curve at low angles of attack is essentially a straight line at low Mach numbers. Also it intersects the zero lift axis at a zero lift angle depending on the basic design of the airfoil. Since the model used here is symmetrical, and was machined to very close tolerances ($\pm$ .001 inch), we know that the angle of zero lift is zero degrees. Thus, the C_L vs. α curve must pass through the origin. Hence it is concluded that the authors of Ref. 1 should have drawn straight lines through their points and shifted these lines so that they passed through the origin. This procedure was followed in the present investigation and gave very good results. Instead of following the above procedure, the authors of Ref. 1 corrected their angle of attack values by comparing their experimental lift curves with theoretical lift curves. They attributed part of the discrepancy in lift curves to angle of attack errors. Consequently, in order to verify their results in the present investigation, the optical system for measuring angle of attack was installed. The angle increments of this investigation are known to be correct within $\pm$ 0.030 degrees. On comparing the lift curves obtained with the experimental points of Ref. 1 it was found that their angle increments turned out to be quite accurate.

Data taken from Fig. 7), Ref. 1 and replotted in Fig. 8 (this report) with results obtained in this investigation shows that a straight line fits their points very well. However, their lift values are low for a given angle of attack. This is attributed to incomplete correction of the lift coefficient for the flow conditions in the wind tunnel.

Further research into the matter of measurement of lift in a two-dimensional wind tunnel was carried out in connection with this investigation and that of Ref. 2. From this research it was found that the lift measured is subject to several wind tunnel effects which reduce the lift coefficient by a considerable degree. All these effects were collected and values computed for this particular wind tunnel, see Ref. 2. When these corrections are applied to the test data the results agree very well with theoretical values expected for this airfoil.

In discussing the results of the primary problem, namely, the effects of the sudden extension of the dive recovery flap, certain pertinent facts can be pointed out by an examination of the experimental curves obtained (Fig. 9-13 inclusive). A consideration of Figs. 9-11 shows that for the flap at 15% chord location the maximum lift coefficients reached, i.e., $C_{L_{crit.}}$, for the curves of flap popped lie above those for the curves of flap set. For the 30% chord location of the flap, the $C_{L_{crit}}$ values of the curves for flap popped are not consistently above or below the values for flap set. At the 40% chord location of the flap the values of $C_{L_{crit}}$ for flap popped

lie below those for flap set in every case. The fact that the effects of the flap popped as compared to flap set changed in character as the flap was moved aft is not considered to indicate that the tests gave inconsistent results. Rather, it shows that the results obtained from such an aerodynamic device depend on both the method of operation of the device and the configuration or proportions of the test model.

All the curves for a given flap location appear to have about the same characteristic shape above a Mach number of 0.6. It is believed that the points above $M = 0.6$ are more accurate because the readings of the multiple tube mercury manometer at these Mach numbers are subject to a smaller percentage of error because of the greater pressure differences.

In the figures discussed above (Figs. 9-11), the Mach number corresponding to $C_{L_{crit}}$ appears to decrease with increasing angle of attack.

An examination of Fig. 12 shows that $C_{L_{crit}}$ increases for both the popped and set conditions as the flap is moved aft. At each flap position the curves are practically parallel, thus indicating that $\Delta C_{L_{crit}}$ is fairly constant with change in angle of attack. A consideration of the change of slope of the curves with increasing angle of attack indicates that the curves may approach a zero slope at some angle of attack above three degrees.

From Fig. 13 it is evident that $\Delta C_{L_{crit}}$ changes sign as the flap is moved aft, and that the curve of $\Delta C_{L_{crit}}$ passes through zero at a flap position of about 30% chord.

A consideration of the schlieren photographs, Figs. 14-16 inclusive, shows that in general there is no appreciable difference in flow pattern and shock wave formations between flap set and popped at the various chord locations. The only apparent difference in shock formation is seen in Fig. 14 at a Mach number of about 0.71. On the model with the flap set down a medium sized turbulent shock is located on the upper surface at about 55% of chord. On the model with flap popped a smaller shock in the same position as above is observed, and in addition several laminar shock waves are seen which are not discernable on the model with flap set. In Fig. 14 with the flap set down at a Mach number of 0.56 a small shock is observed on the upper surface near the leading edge. Thus the flow has already reached the critical Mach number. As the Mach number increases the supersonic zone increases in extent, and more laminar shocks become visible. At a Mach number of 0.71 a turbulent shock is observed at about 55% chord.

Comparison of the model with the flap at 30% chord to that with the flap at 15% chord, at a Mach number of 0.56, shows that the laminar shock formation in the case of the 30% chord location is more extensive, and shocks have apparently increased in strength. As the Mach number is increased the extent of laminar shocks grows more rapidly than it did on the model with the flap at 15% chord, and a turbulent shock becomes visible at a Mach number of 0.693.

On the model with the flap located at 45% chord the shock wave formation and flow patterns are essentially the same as those on the model with the flap located at 30% chord, with the following exceptions. On the model with the flap located at 45% chord the shock formation is stronger; and an envelope of Mach waves has appeared at $M = 0.$[illegible] and at $M = 0.714$, which did not appear in the case of the model with the flap at 30% chord.

A comparison of the pictures (Figs. 14-16 incl.) shows that the effect of moving the flap from 15% to 30% chord is more than the effect of moving the flap from 30% to 45% chord. At a Mach number of 0.648 on the model with the flap located at 30% chord there is visible a nearly normal laminar shock which is also visible with slightly increased strength on the model with the flap at 45% chord. This shock is not at all visible on the model with the flap at 15% chord. At a Mach number of 0.[illegible] strong turbulent shocks of approximately equal strength are observed on both the model with the flap at 30% chord and that with the flap at 45% chord, but not on the model with the flap at 15% chord. Also, the supersonic zone is more extensive on the models with flaps located at the 30% and 45% chord positions than it is on the model with the flap located at 15% chord.

No shock waves are observed on the lower surface of the model for any test condition. This indicates that the flow past the lower surface remained subsonic for the flap either popped or set down at any of the three chord locations tested.

It is considered that the pictures made with the flap suddenly extended represent the flow conditions after they have become steady,

since the average interval of time between the popping of the flap and the taking of the picture was of the order of 15 seconds. The flow conditions for the flap set down are considered to be stable as the speed was built up from zero to a given Mach number with the flap fixed in position.

CONCLUSIONS

1. With an accurate angle of attack measuring system the lift tests of the symmetrical airfoil model with no flap gave a straight line lift curve at low angles of attack.

2. The test results show that the aerodynamic effects of a flap popped, or suddenly extended, as compared to a flap that is set in position are different in the transonic range. The flap popped at the 15% chord location gave approximately 8.0% higher lift than the flap set. At the 30% chord location there was no distinguishable difference. At the 45% chord location the popped flap gave approximately 3.0% less lift than did the set flap.

3. Wind tunnel test models which employ dive recovery flaps set in position and which are used to extend test values to full scale design may give misleading results. Whenever model test results are to be used to produce design information to be incorporated in full scale aircraft, the dive recovery flaps on the model should be equivalent to those on the full scale airplane not only as to dimensional proportions but also as to method and timing of operation, if a high degree of accuracy is desired.

4. The extension of a dive recovery flap on an airfoil increases the circulation and therefore the lift and the peak of the pressure distribution curve. Hence, the critical Mach number is lowered.

RECOMMENDATIONS

In order to verify and further extend the results obtained in this investigation, a more detailed study should be made of the effects on the flow conditions resulting from the flap being set and popped at one chord location. If this is done, the following specific recommendations are made:

(a) A study of the boundary layer by means of schlieren pictures and a wake survey should furnish additional information of value.

(b) It is recommended that the height of the multiple tube manometer be increased to allow for large and rapid pressure variations which occur when a device such as a dive recovery flap is suddenly extended. This change would permit the use of a liquid with a specific gravity of 2.0 or less throughout the entire range of an investigation, thereby providing increased accuracy.

(c) The mechanical difficulties encountered and suggested improvements in equipment which are given in Ref. 2 apply also to this investigation.

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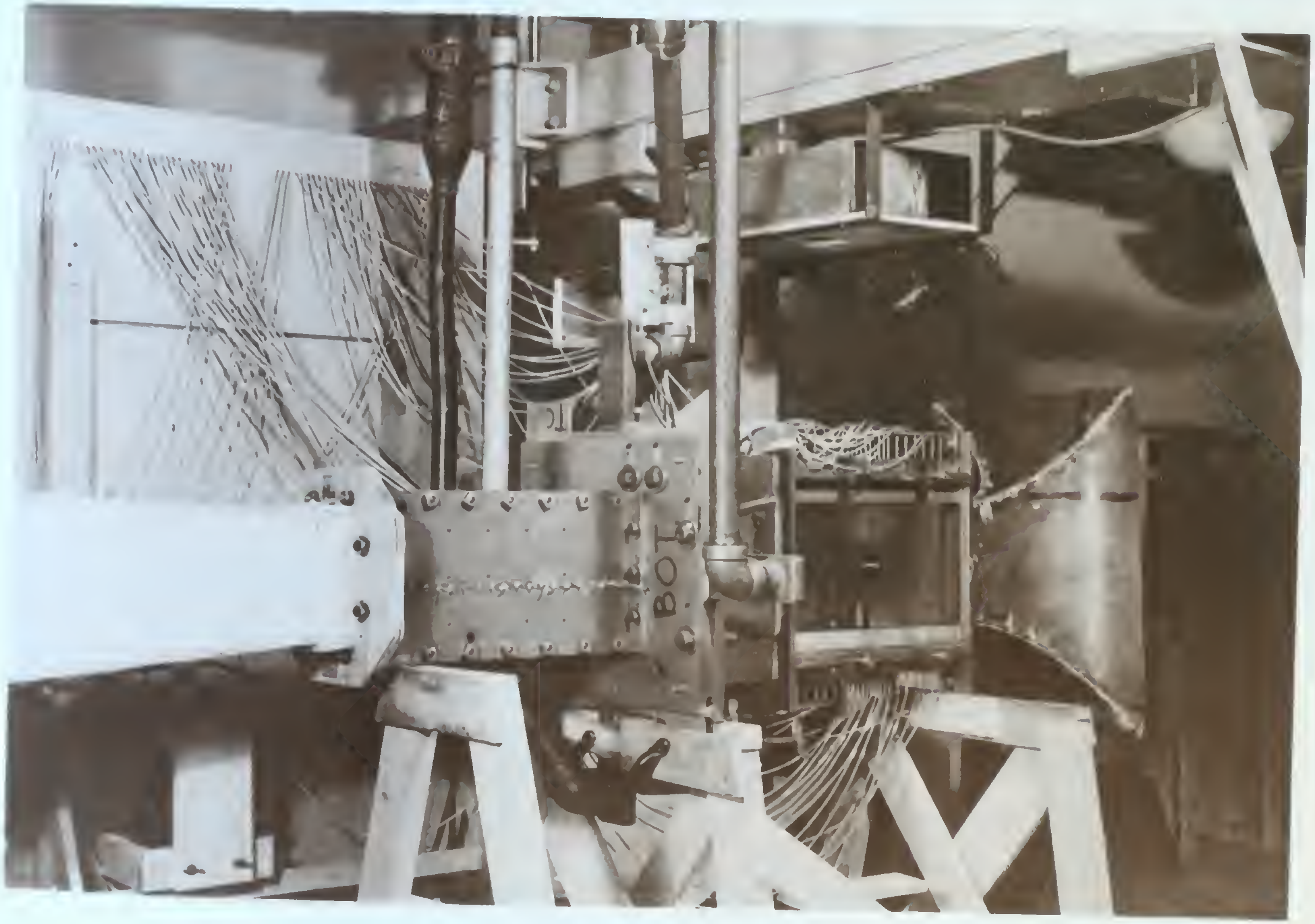
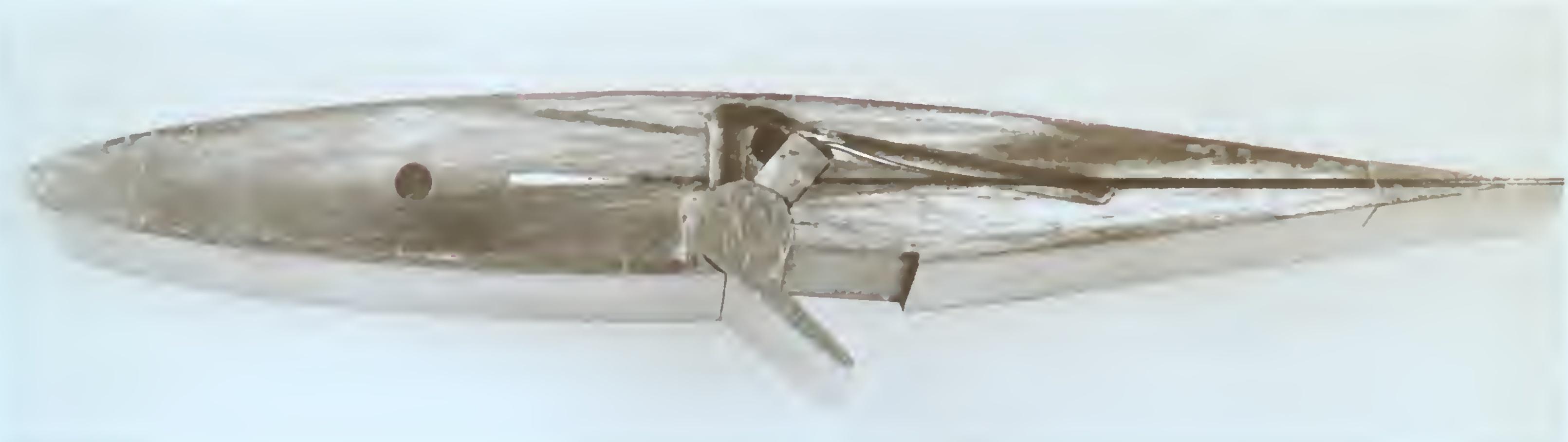


Fig. 1

Wind Tunnel

Flap retracted



Flap extended

Fig. 2

Model of NACA Airfoil 65, 1-012
Equipped with Dive Recovery Flap

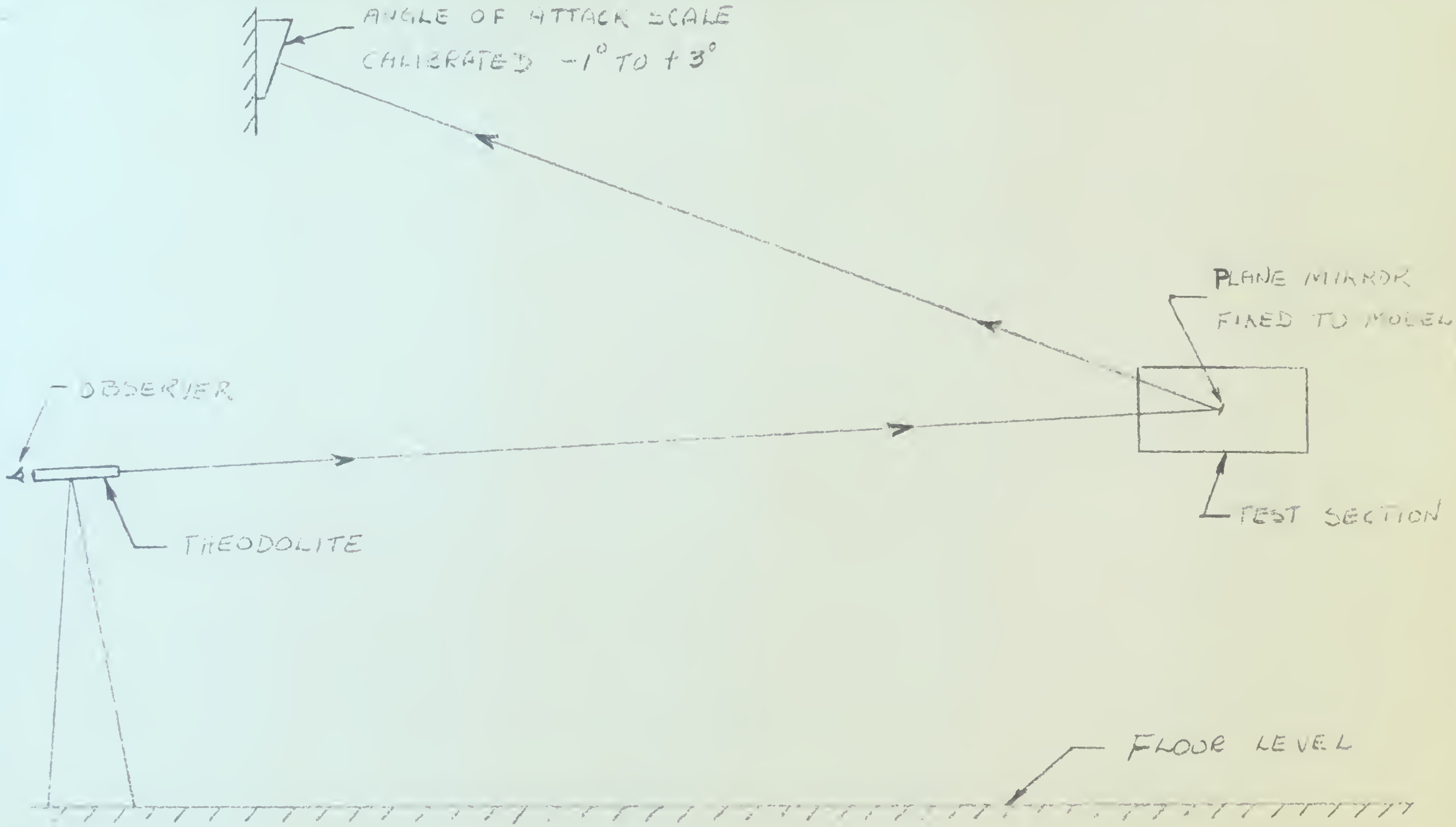


FIG. 3

SCHEMATIC DIAGRAM OF ANGLE OF ATTACK SYSTEM

SCALE 1" - 20'

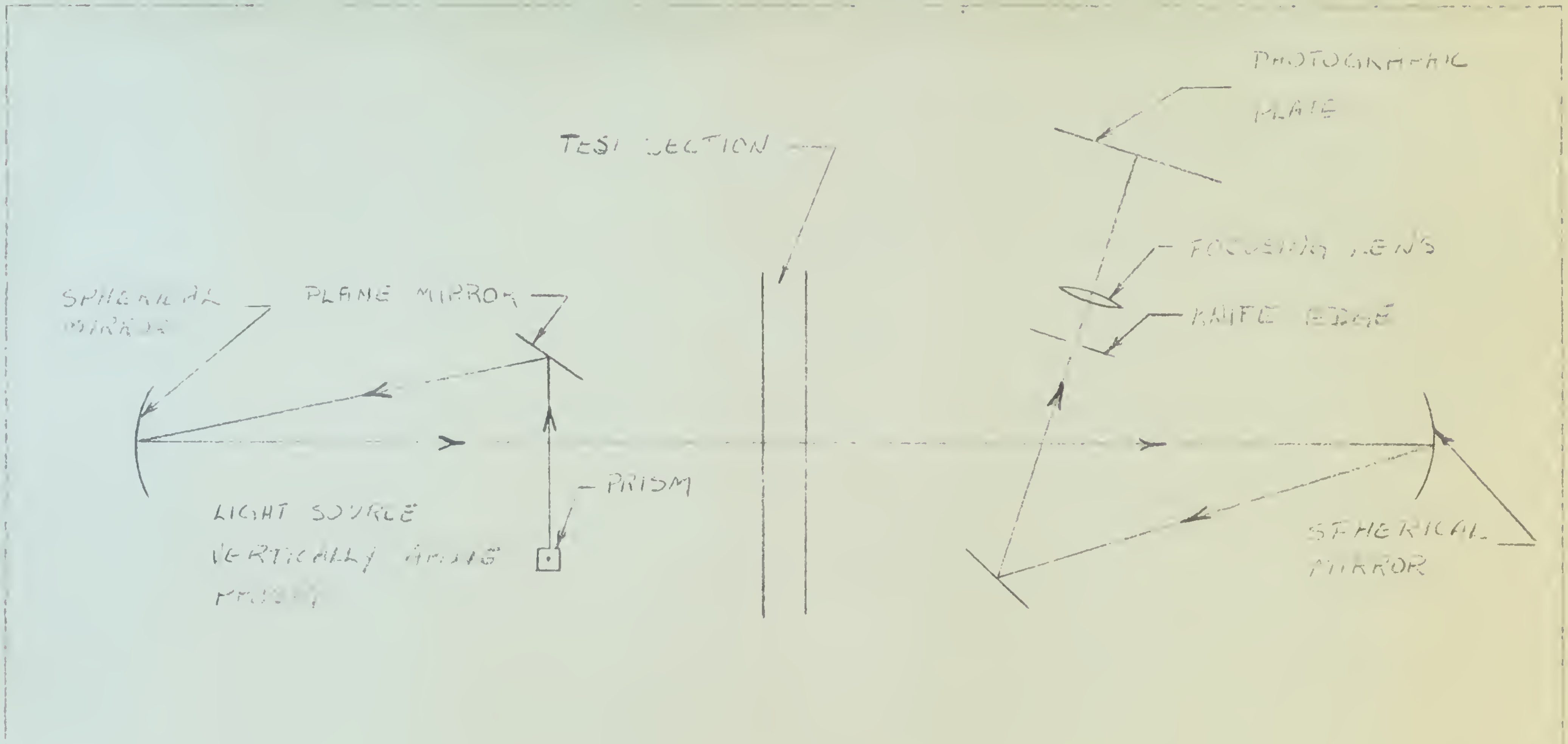


Fig. 4
Schematic Diagram
of Schlieren Apparatus

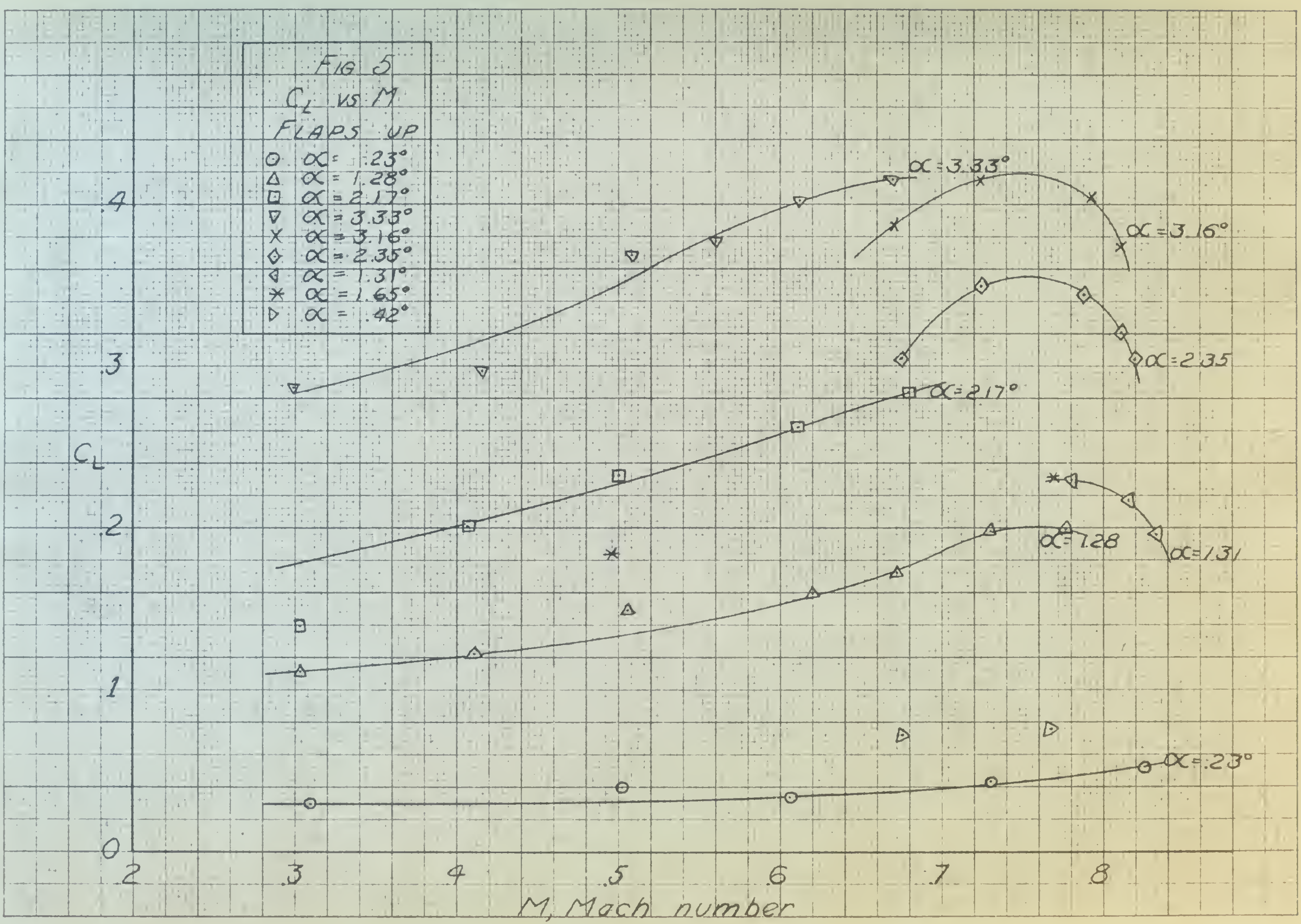
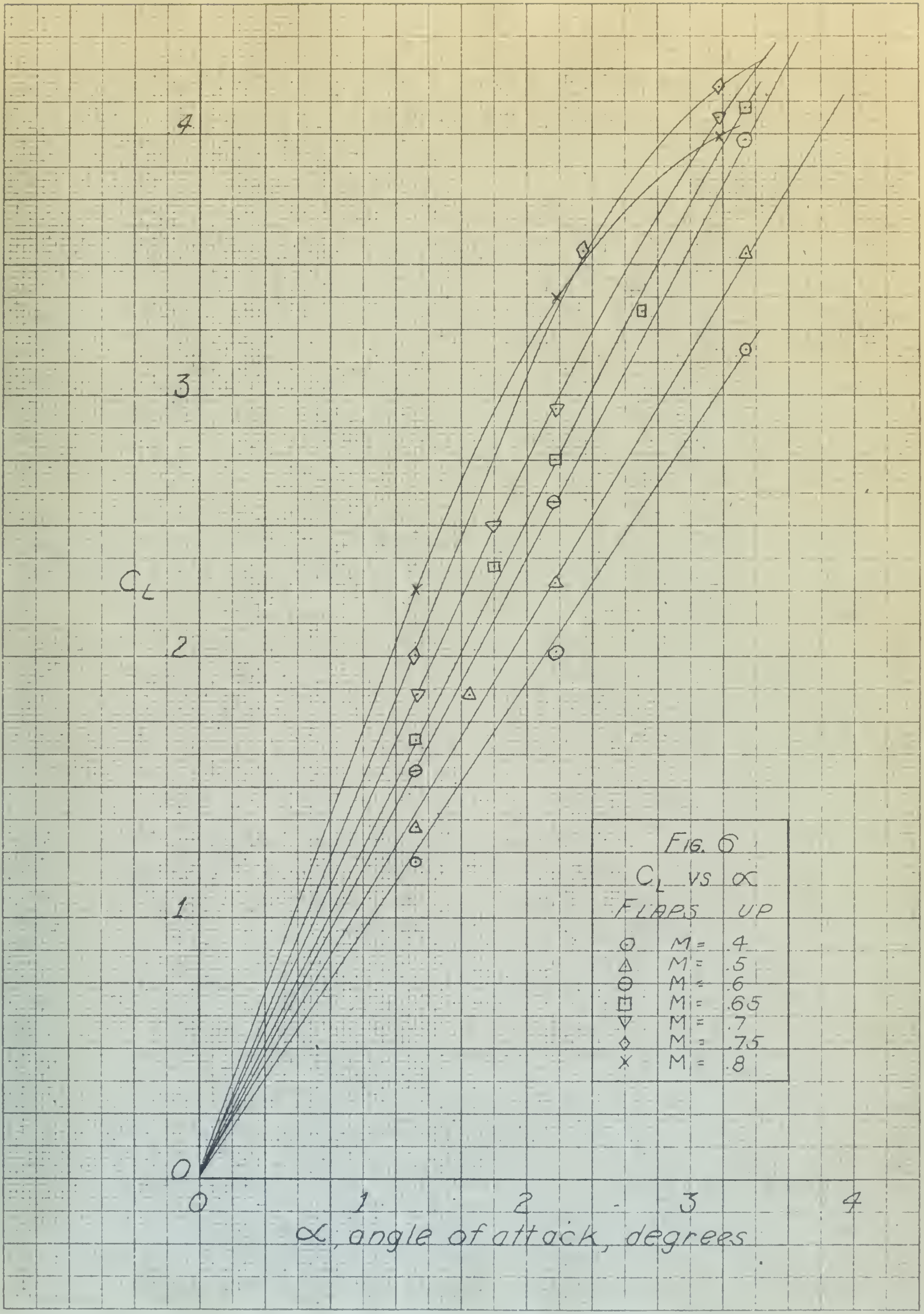


FIG 5
C_L VS M
FLAPS UP
⊙ α = .23°
△ α = 1.28°
⊡ α = 2.17°
▽ α = 3.33°
X α = 3.16°
◇ α = 2.35°
◁ α = 1.31°
✳ α = 1.65°
▷ α = .42°
α = 3.33°
α = 3.16°
α = 2.35
α = 2.17°
α = 1.28
α = 1.31
α = .23°
C_L
0
1
2
3
4
.2
.3
.4
.5
.6
.7
.8
M, Mach number


4
3
2
1
0
C_L
0
1
2
3
4
α, angle of attack, degrees
Fig. 6
C_L vs α
Flaps up
M = .4
M = .5
M = .6
M = .65
M = .7
M = .75
M = .8

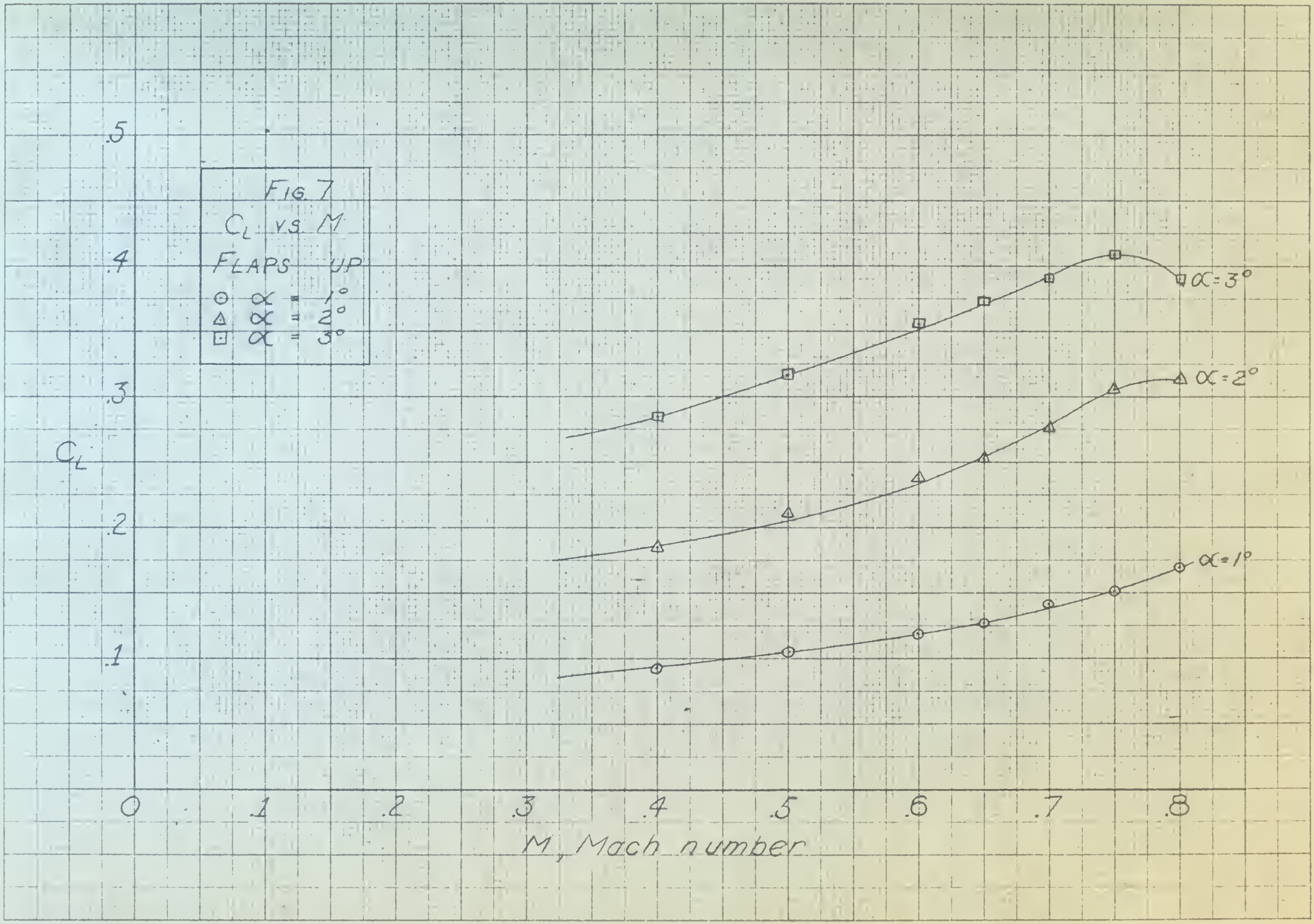
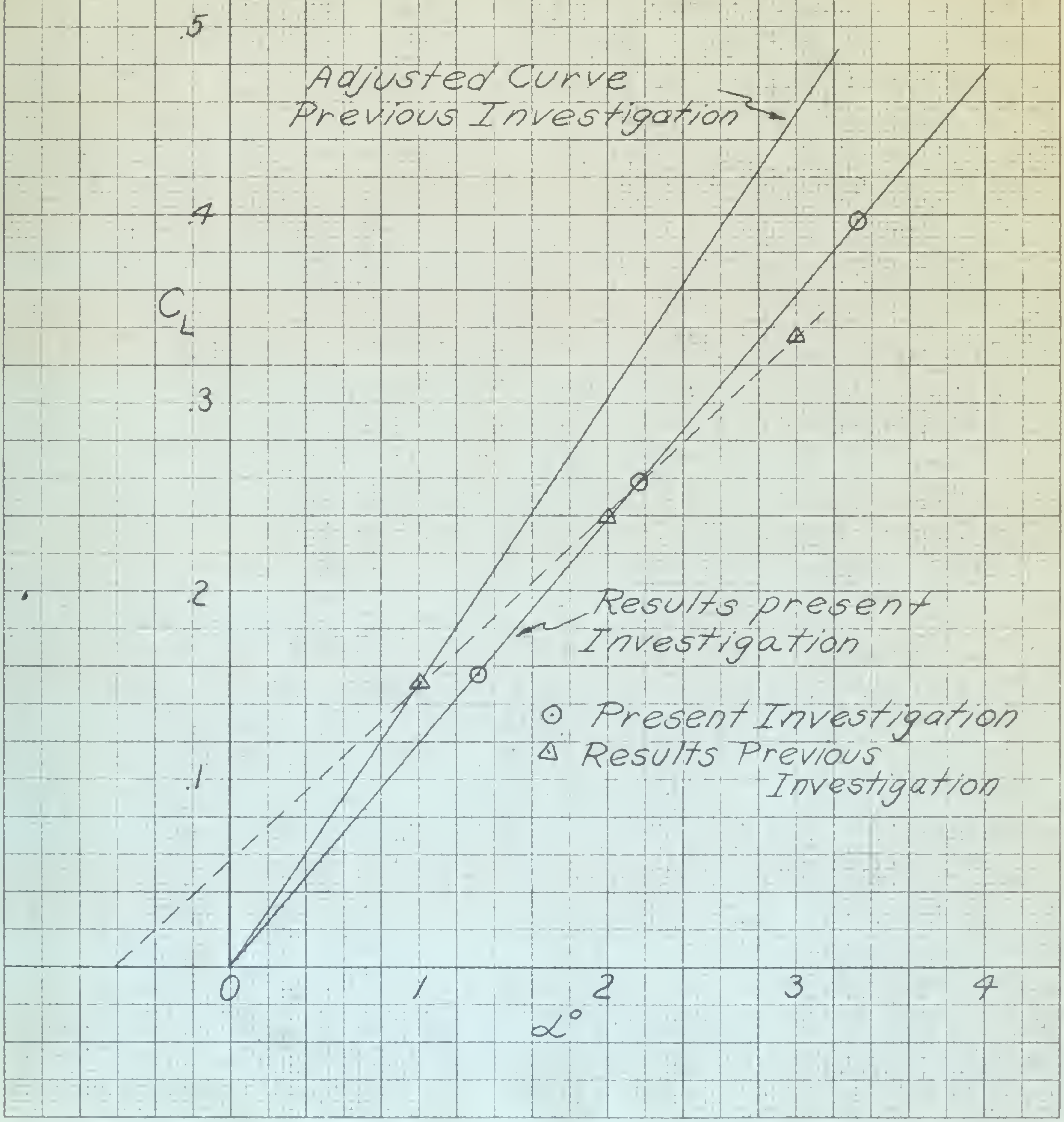


FIG 7
C_L vs M
FLAPS UP
⊙ α = 1°
△ α = 2°
⊡ α = 3°
α = 3°
α = 2°
α = 1°
C_L
.5
.4
.3
.2
.1
0
.1
.2
.3
.4
.5
.6
.7
.8
M, Mach number

FIG. 8

COMPARISON OF LIFT CURVE
WITH FIG. 70; REF. 1; M=0.6



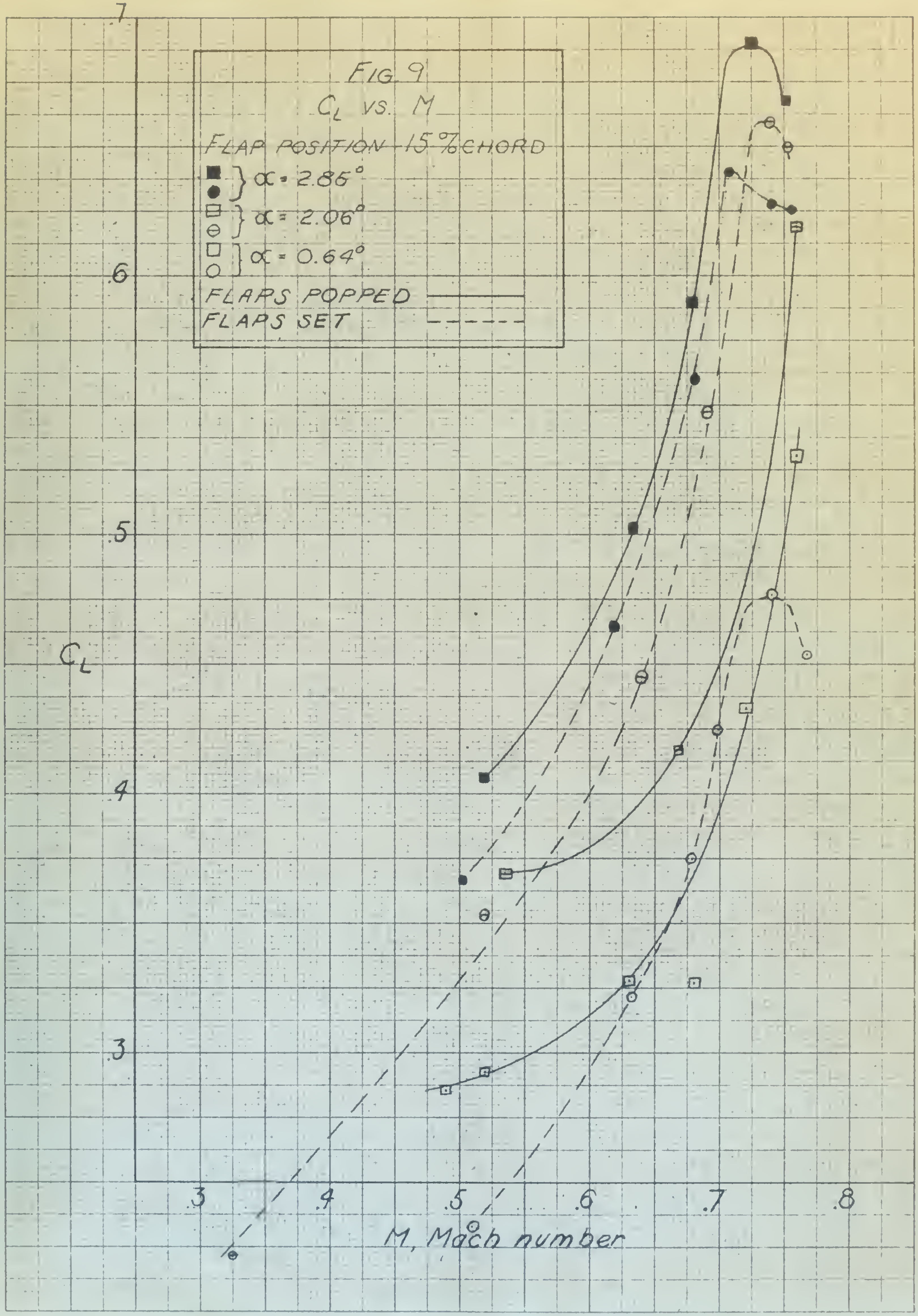


FIG 9
C_L VS. M
FLAP POSITION 15% CHORD
$\alpha = 2.86°$
$\alpha = 2.06°$
$\alpha = 0.64°$
FLAPS POPPED
FLAPS SET
.7
.6
.5
.4
.3
C_L
.3
.4
.5
.6
.7
.8
M, Mach number

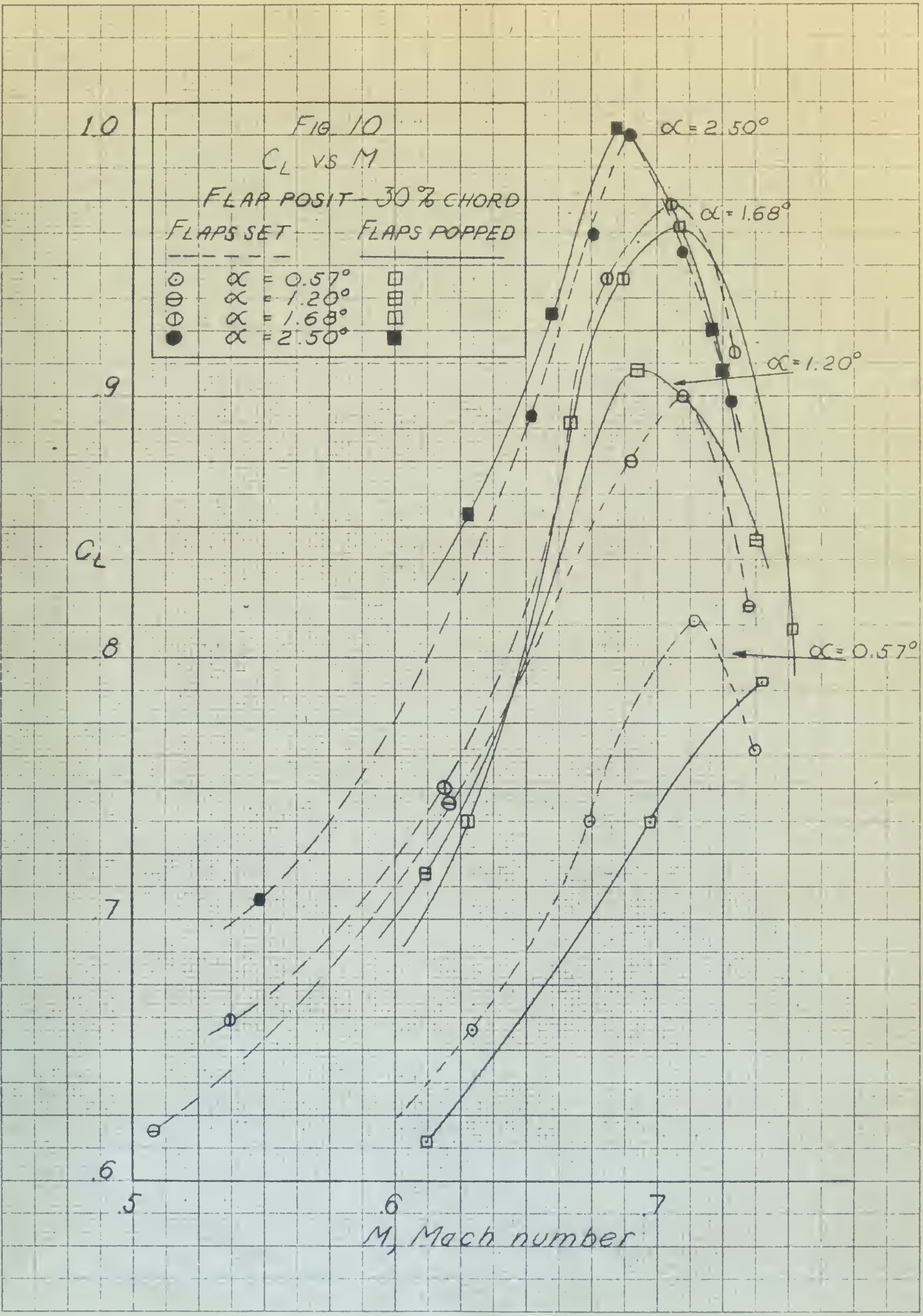


FIG 10
C_L VS M
FLAP POSIT - 30% CHORD
FLAPS SET
FLAPS POPPED
α = 0.57°
α = 1.20°
α = 1.68°
α = 2.50°
α = 2.50°
α = 1.68°
α = 1.20°
α = 0.57°
1.0
.9
.8
.7
.6
C_L
.5
.6
.7
M, Mach number

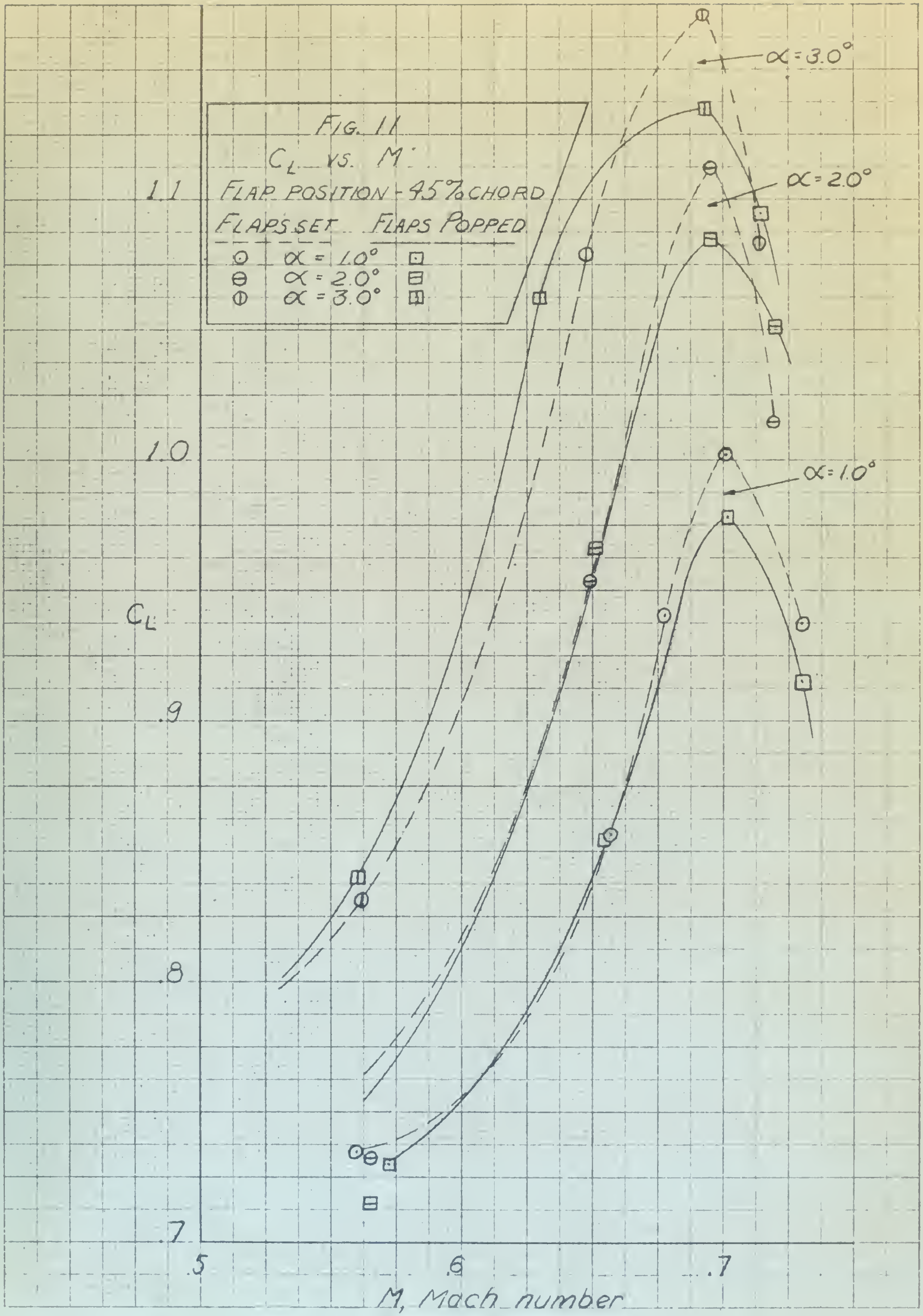


FIG. 11
C_L VS. M
FLAP POSITION - 45% CHORD
FLAPS SET
FLAPS POPPED
α = 1.0°
α = 2.0°
α = 3.0°
α = 3.0°
α = 2.0°
α = 1.0°
1.1
1.0
.9
.8
.7
C_L
.5
.6
.7
M, Mach number

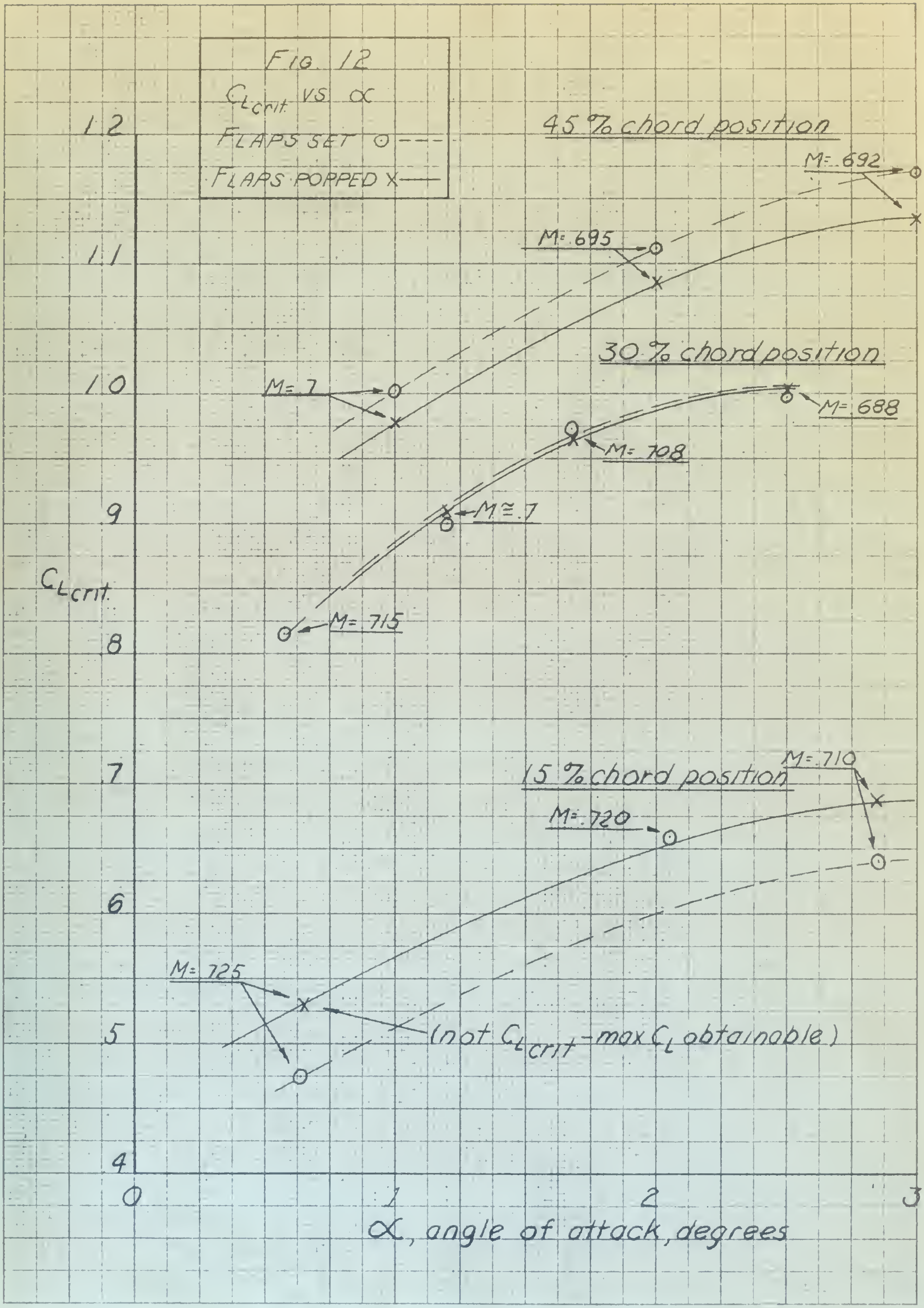


Fig. 12
$C_{L_{crit}}$ vs α
FLAPS SET ⊙ - - -
FLAPS POPPED × ——
45 % chord position
M=.692
M=.695
M=.7
30 % chord position
M=.688
M=.708
M≅.7
M=.715
15 % chord position
M=.710
M=.720
M=.725
(not $C_{L_{crit}}$ - max C_L obtainable)
$C_{L_{crit}}$
1.2
1.1
1.0
.9
.8
.7
.6
.5
.4
0
1
2
3
α, angle of attack, degrees

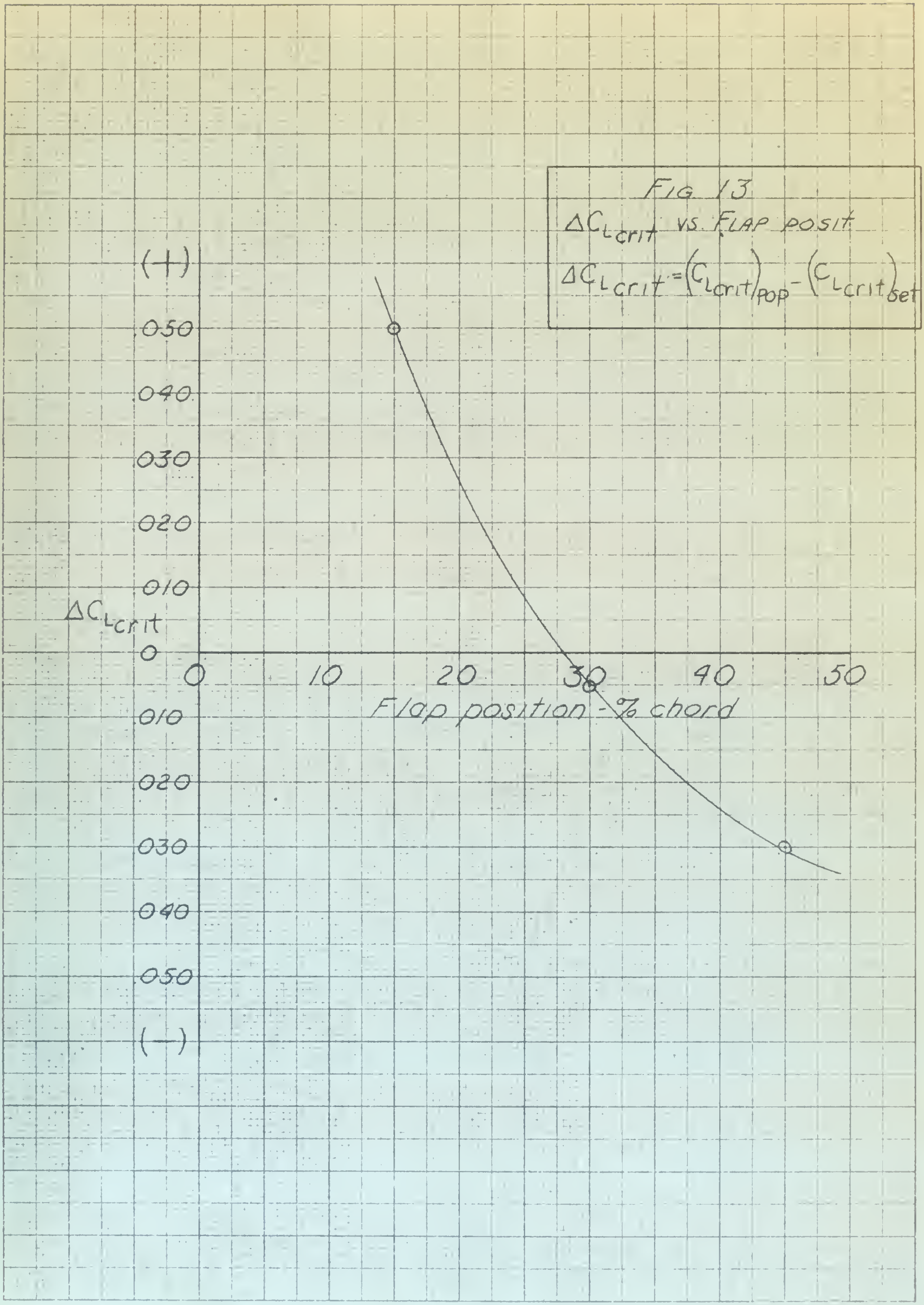


FIG. 13

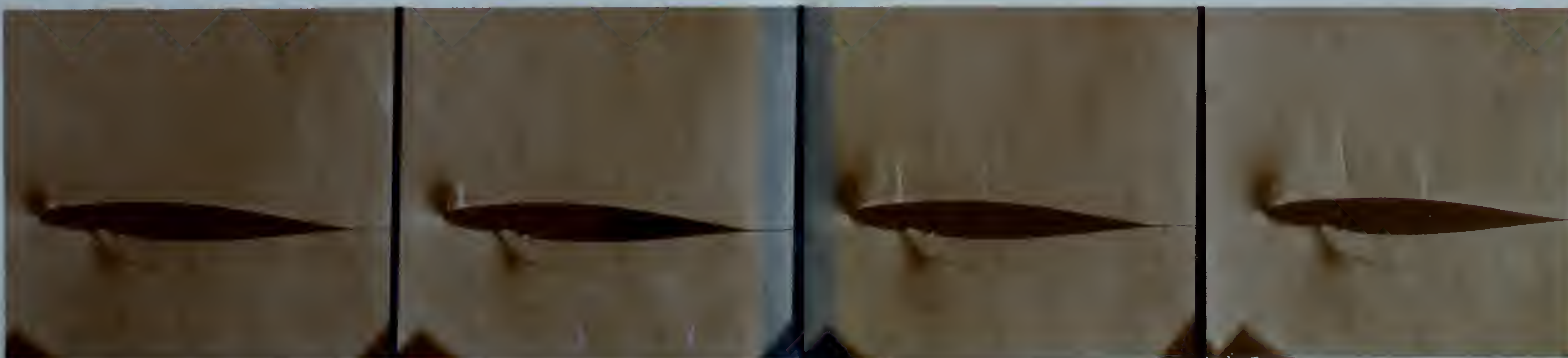
$\Delta C_{L_{crit}}$ vs. FLAP POSIT.

$\Delta C_{L_{crit}} = \left(C_{L_{crit}}\right)_{POP} - \left(C_{L_{crit}}\right)_{set}$

Fig. 14

M = .561 Flaps Set M = .648

M = .693 Flaps Set M = .714



M = .558 Flaps Popped M = .65

M = .69 Flaps Popped M = .71

Fig. 14

Flap Location - 15% Chord
$\alpha = 3^{\circ}$

Fig. 15

M = .561 Flaps Set M = .648

M = .693 Flaps Set M = .714

M = .558 Flaps Popped M = .65

M = .69 Flaps Popped M = .71

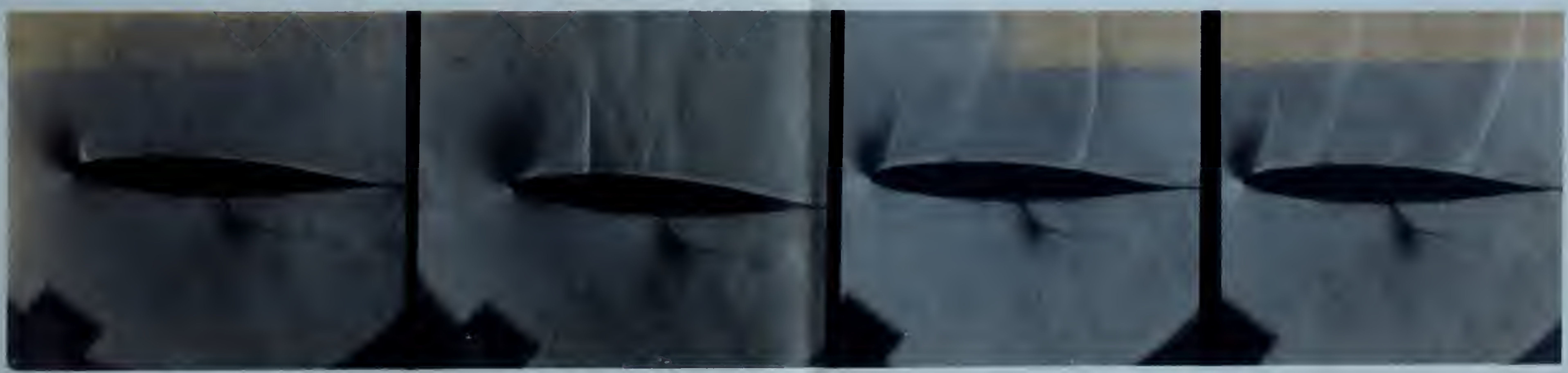
Fig. 15

Flap Location at 30% Chord
$\alpha = 3^{\circ}$

Fig. 16

M = [redacted] .561 Flaps Set M = .648

M = .693 Flaps Set M = [redacted] .714



M = .558 Flaps Popped M = .65

M = .69 Flaps Popped M = .71

Fig. 16

Flap Location at 45% Chord
$\alpha = 3^{\circ}$

DATE DUE

Parker

An investigation of the effects of the sudden extension of a dive-recovery flap on the aero-dynamic characteristics of

of a symmetrical airfoil in two dimensional flow